

SYSTEM LEVEL MECHANICAL TESTING OF THE CLEMENTINE SPACECRAFT

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Abstract

This paper discusses the system level structural testing that was performed to qualify the Clementine Spacecraft for flight. These tests included spin balance, combined acoustic and axial random vibration, lateral random vibration, quasi-static loads, pyrotechnic shock, modal survey and on-orbit jitter simulation. Some innovative aspects of this effort were:

- The simultaneously combined acoustic and random vibration test
- The mass loaded interface modal survey test
- The techniques used to assess how operating on board mechanisms and thrusters affect sensor vision

Definitions and Acronyms

BMDO	Ballistic Missile Defense Organization - Co-sponsor of the Clementine Program
ELV	Expendable Launch Vehicle
EM	Engineering Model - High fidelity Prototype model of the Clementine Flight Spacecraft.
ISA	Interstage Adapter - Composite truncated cone adapter that interfaced the Solid Rocket Motor to the Clementine Spacecraft
LLNL	Lawrence Livermore National Laboratory - Supplied the lightweight sensors and cameras that were used for lunar mapping
NASA	National Aeronautics and Space Administration - Co-sponsor of the Clementine program, also supplied the use of the Deep Space Network for communications
NCST	Naval Center for Space Technology at the Naval Research Laboratory
NRL	Naval Research Lab
PA	Payload Adapter - aluminum truncated cone adapter that interfaced the Space Vehicle to the Titan IIG Launch Vehicle
Payload	The payload consists of the Payload Adapter (PA), the Interstage Adapter (ISA), Solid Rocket Motor (SRM) and Clementine Spacecraft.
SC	Clementine Spacecraft
SV	Space Vehicle - Assembly consisting of the Interstage Adapter, Solid Rocket Motor and Clementine Spacecraft

Introduction

The Clementine Spacecraft, designed, built, and integrated by the Naval Center for Space Technology at the Naval Research Laboratory, was launched on January 25, 1994 on a Titan II ELV from Vandenberg Air Force Base. This was the culmination of a 22-month odyssey from concept to launch. The Clementine program was jointly sponsored by the Ballistic Missile Defense Organization (BMDO) and the National Aeronautics and Space Administration (NASA). Using an array of light weight sensors and cameras, supplied by Lawrence Livermore National Laboratory (LLNL), it successfully completed the most extensive mapping of the moon to date.

There were three fundamental problems that faced the Clementine program related to system level testing due to the "fast-track" approach that was required; 1) there was no tolerance for major schedule delays, 2) financial and schedule constraints precluded the implementation of a full qualification program, and 3) there were uncertainties with regards to the engineering issues associated with the new technologies. As a result, in order to provide an acceptable combination of time, money and risk, a hybrid qualification/protoflight approach was adopted.

Two identical spacecraft structures were fabricated. The first was designated as the Engineering Model (EM) and the second was identified as the Flight Model. The EM was assembled during the period of January 1993 to May 1993. System level testing on the EM was conducted from June 1993 through August 1993. This provided data early in the program that proved invaluable in support of the design development. The EM was also used for verification of the basic design, the systems engineering, and was a pathfinder for the flight unit. The EM system testing schedule was developed and performed in support of the flight spacecraft build. Assembly of the flight spacecraft began in May 1993 and was completed in August 1993. Integration took place from August 1993 to October 1993. System level testing at NRL started at the beginning of November 1993 and concluded at the end of December 1993. Clementine was shipped to Vandenberg AFB, CA, on December 30, 1993 for launch preparation where final system integration and testing took place.

Mission Background

Conceived as a implement for testing advanced lightweight space-based technologies, conceptual design of the Clementine mission began in the early spring of 1992. The mission objectives for Clementine were to demonstrate the latest in BMDO developed space-based imaging sensors and advanced lightweight component technologies on a long duration flight. This was to be achieved by employing as much existing BMDO technology as possible to perform a lunar mapping as well as an asteroid encounter. A "fast track" program philosophy was required to meet an early 1994 launch.

Utilizing targets such as the moon, the stars, and the spacecraft interstage assembly (after separation), the performance of these technologies were demonstrated. As an alternate objective to the primary military mission, but one of scientific importance to the international civilian community, was to digitally map the lunar surface. Clementine entered lunar orbit on February 19, 1994. It gathered the most comprehensive lunar multispectral image collection to date, collecting over 1.8 million pictures, by May 3, 1994. The imaging of the lunar surface has been made available to the planetary science community. Figures 1 and 2 illustrate some of the images that were captured.

A secondary objective was developed during the mission design that utilized the spacecraft interstage assembly as a radiation and particle detection experiment. After separation from the spacecraft, the interstage assembly remained in a highly elliptical earth orbit for several months collecting valuable data at altitudes where opportunity for investigation has not been available.

The Clementine program successfully demonstrated the capability to produce, implement, and operate spacecraft of the "faster, cheaper, better" variety through cooperative efforts between national laboratories, DOD, NASA, Industry, and international space organizations.

Mission Scenario (and Resulting Environments)

- The Clementine Spacecraft was launched on a Titan IIG ELV from Vandenberg AFB into a low earth staging orbit (LEO) on January 25, 1994. (Launch environments were a combined 11.0 Gpk axial and 3.5 Gpk lateral quasi-static loads as well as vibro-acoustic loads.)
- After achieving LEO, the launch Clementine payload separation was initiated by the Titan II ELV. This separation involved releasing the marmon clamp that mated the Clementine Space Vehicle to the Payload Adapter. (Pyroshock event)
- After several days in LEO the payload was spun up to 60 RPM using the Spacecraft's attitude control thrusters. At this point, the Space Vehicle was injected into a trans-lunar trajectory using the Star 37FM SRM which was mounted in the Interstage Adapter. (12.5 Gpk quasi-static axial acceleration)
- Next the attitude control system thrusters were activated to stop the Space Vehicle Spin and the solar arrays were deployed by releasing the frange bolts. (Pyroshock event)
- The Clementine Spacecraft then initiated separation from the Interstage Adapter and SRM by firing eight separation nuts. (Pyroshock event)
- The expended SRM/ISA was left in a highly elliptical orbit while the Clementine Spacecraft was inserted into a stable lunar polar orbit using the Spacecraft 110 lbf. delta V thruster.
- Lunar mapping operations using the Clementine SC's mission sensors over a period of approximately two months.

Clementine Design

Referring to figures 3 and 4, the Clementine payload was designed to provide structural, mechanical and electrical interfaces with the Titan IIG ELV. The payload consisted of:

- The Payload Adapter (PA) - An aluminum truncated cone with riveted forward and aft interface rings which connected the Clementine Space Vehicle to the Titan IIG Launch Vehicle. The PA weighed approximately 85 lbs.
- The Interstage Adapter (ISA) - A composite truncated cone with forward and aft ring which were riveted and bonded to the shell. The ISA housed the Solid Rocket Motor and interfaced it to the Clementine Spacecraft. The ISA weighed approximately 80 lbs.
- The Solid Rocket Motor (SRM) - The SRM provided 12,000 lbf. of thrust for lunar orbit insertion. The SRM bolted to the aft ring of the ISA at the Space Vehicle separation plane. The SRM weighed approximately 2500 lbs. and was by far the heaviest component.
- The Clementine Spacecraft which provided a stable platform and positioning for the optical mission sensors. The Spacecraft's structural components were a machined aluminum subfloor, two decks containing the Reaction Control System and thrusters, eight honeycomb aluminum sheer panels containing the electronic components/boxes and eight longerons attaching the subfloor with the decks and sheer panels. The Spacecraft weighed approximately 490 lbs. unfueled.

Clementine Overall Test Philosophy

The Clementine Satellite best fits in the category of a Class C Space Vehicle. DOD-HDBK-343 describes Class C Satellites as a medium to high risk effort, single string design, small size and low to medium complexity. In order to provide the best mix of mechanical reliability and low cost, a combination Qualification / Protoflight test philosophy was adopted which had the following characteristics:

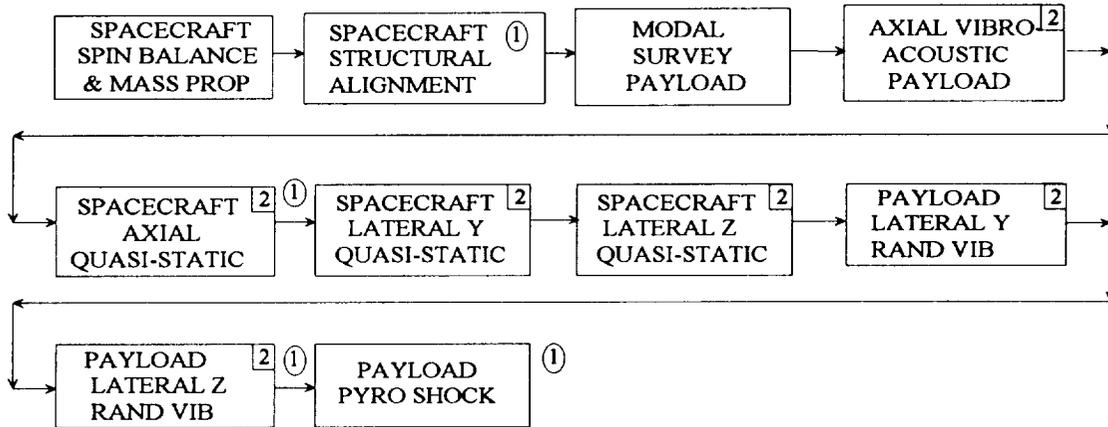
- Test environments and loads were based on using a Titan IIG Expendable Launch Vehicle to get the Space Vehicle into low earth orbit, and a Thiokol Star 37FM Solid Rocket Motor for putting the Spacecraft into lunar orbit.
- Separate Engineering Model and Flight Spacecrafts were built.
- Qualification test levels were defined as maximum expected flight levels plus 6 dB for two minutes duration. Protoflight test levels were defined as maximum expected flight levels plus 3 dB for one minute duration.
- The Engineering Model Spacecraft (EM) was tested to qualification levels.
- The Flight Spacecraft was acceptance tested to protoflight levels.
- Components with only flight units available underwent protoflight testing while components with both operational prototypes and flight units had qualification testing on the prototype and protoflight testing on the flight unit.
- Levels for loads testing were equal to 1.05 times expected flight loads. The structure was designed for yield at a minimum of 1.10 times flight loads.
- Low level random signature tests were performed before and after any system level test that input significant loads into the structure. Transfer functions from these pre and post test signatures were compared to see if any structural failure or degradation had occurred.

Test Flow

The test flows for both the EM and Flight Spacecraft are shown below. The goal of the EM test program was to qualify the Clementine design and provide a pathfinder for future Flight Spacecraft testing. The goal of the Flight Spacecraft test flow was to verify workmanship and acceptability of Clementine in its entirety for flight.

Engineering Model Test Flow

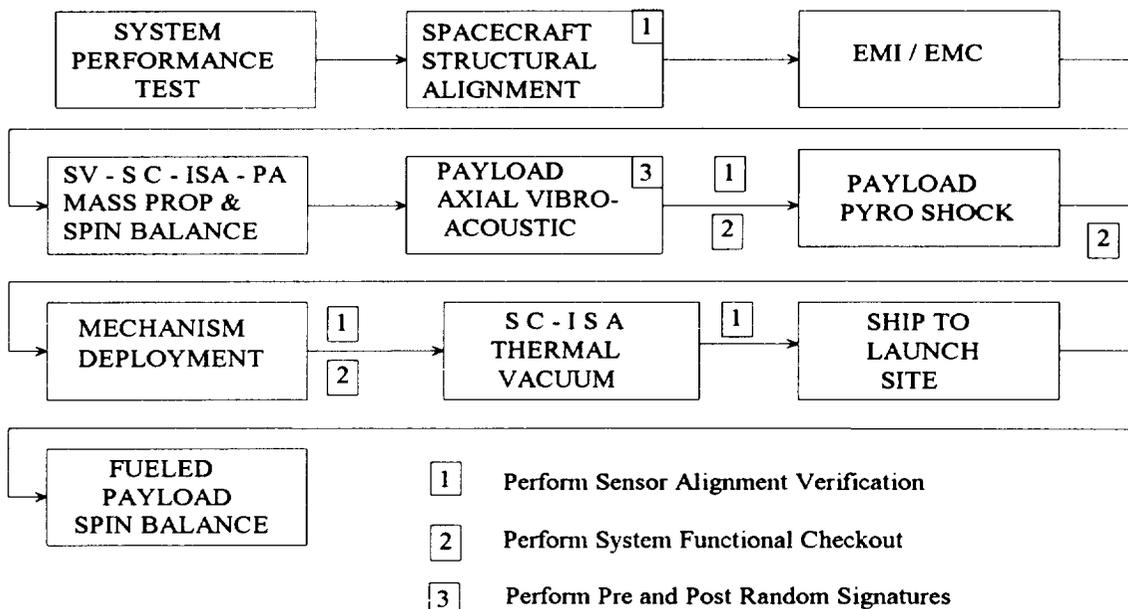
The Engineering Model (EM) was a high fidelity, non-flight, mass and stiffness prototype of the Clementine Spacecraft which was tested to qualify Clementine's design. It contained mass simulators for all components as well as dummy solar arrays. For tests performed on the Space Vehicle (S/V) configuration, a mass simulated SRM and high fidelity prototypes of ISA and PLA were used instead of the flight items. The following flow chart illustrates the Engineering Model development test flow.



- ① PERFORM SENSOR ALIGNMENT VERIFICATION
- ② PERFORM PRE AND POST STRUCTURAL SIGNATURES

Flight Structure Test Flow

The Flight Clementine Spacecraft in launch configuration was acceptance tested to protoflight levels. The flight PLA and ISA were used for these tests, but for safety reasons the SRM mass simulator was used instead of the flight SRM.



- ① Perform Sensor Alignment Verification
- ② Perform System Functional Checkout
- ③ Perform Pre and Post Random Signatures

Sensor Bench Location Alignment

Angular alignment verification of Clementine's optical sensors to one another was critical to mission success. Sensors were aligned before testing began and alignment was rechecked periodically during the test flow to determine sensitivity to dynamic and thermal loads. Translational alignment was not nearly as critical as angular alignment because the sensors were viewing subjects that were hundreds of miles away. The Alignments were also measured to provide a check on the overall structure integrity and thruster pointing. Similar alignments were done on both the EM and the Flight Spacecraft.

First, the centerlines of the sensors were aligned to each other, then the alignment of one of the sensors was measured to the reference axis of the Spacecraft. The alignment of the centerlines of the delta V thruster, attitude control thrusters, and Inertial Measuring Units were then also measured to the reference axis of the Spacecraft. Knowledge was required to within .005 degrees on all measurements. All measurements were periodically verified as shown in the test flow.

Spin Balance and Mass Properties

Due to the 3 axis pointing requirements of the spacecraft during lunar orbit and 60 RPM spin during trans lunar flight, balancing and knowledge of mass properties were critical on Clementine.

Engineering Model Structure

Several spin balance tests and mass properties measurements were performed on the Clementine EM as a pathfinder for the flight structure. Spin balance fixtures were balanced separately first. The first test was done on the EM Spacecraft alone (Spacecraft Configuration), mounted on the spin table via the S/V spin fixture and S/C adapter. Balance weights were added to the EM as required. The other spin balance test was done on the EM along with the interstage (ISA) and solid motor simulator (Space Vehicle Configuration). C.G., moments of inertia and products of inertia were measured on the EM S/C using an MOI machine.

Flight Structure at NRL

Engineering Model Spin Balance Pathfinder Activities

Pathfinder activities to prove the ability to accurately mate separate space vehicle components without violating the ACS unbalance condition requirements took place during the months of August 1993 through October 1993. The spacecraft EM and the interstage EM-SRM mass simulator assembly were balanced separately using the NRL spin balance machine. These two components were assembled and the unbalance was measured for the final assembly. Since the flight SRM manufacture was not complete at this time, the SRM mass simulator was configured into a worst case unbalanced condition based on past history of the SRM and contractual agreements with the manufacturer, Thiokol Corporation. The unbalance condition of the interstage EM was measured. The unbalanced SRM mass simulator was then mated to the interstage EM in a worst case configuration by aligning the phase angles of the unbalance properties for the two components. The static and dynamic unbalance of the interstage EM-SRM assembly was then corrected and measured. By mating in a worst case configuration, the allotted weight budget for counterweights was verified.

The spacecraft EM, a structural and mass simulator of the flight spacecraft (stowed wet launch configuration) was also dynamically balanced. The dynamic mode was used because it satisfied both the static requirement during spacecraft 3-axis stabilization and the dynamic requirement during space vehicle spin-stabilization. Static and dynamic unbalance was measured for the final correction. The spacecraft EM and interstage EM-SRM mass simulator were then mated with the use of optical alignment

tooling. The resulting unbalance condition was measured and verified to be within the ACS requirements. Therefore, the following was confirmed: 1) the validity of the spin balance flow, 2) the ability to accurately mate the separate space vehicle components, and 3) the allowed counterweight budget.

Flight Vehicle Spin Balance Processing Operations

During the months of November and December of 1993, the following spin balancing activities took place at Building A-59 of NRL. Using the NRL spin balance machine, both the dry flight spacecraft and the flight interstage-SRM mass simulator assembly were balanced separately. Static and dynamic unbalance was measured and corrected for the dry spacecraft in the stowed launch configuration. The SRM mass simulator representing the unbalance properties of the actual flight SRM was mated to the flight interstage. The unbalance of the SRM mass simulator was previously matched to simulate the actual unbalance of the loaded flight SRM as specified in the Thiokol STAR 37FM Rocket Motor Logbook. The unbalance properties of the SRM mass simulator are based on data collected during flight SRM spin balance operations conducted at NASA Wallops Flight Facility in October 1993. Thiokol Corporation was responsible for the balancing of the flight SRM. Static and dynamic unbalance was also measured and corrected for the flight interstage-SRM mass simulator assembly.

Flight Structure at Vandenberg AFB

Following spin balance operations, both the spacecraft and interstage were shipped to Vandenberg AFB for launch processing. After spacecraft propellant loading and interstage-flight SRM integration at Building 1610 of Vandenberg AFB, the two components were mated together in the space vehicle configuration. Final unbalance measurements of the space vehicle were made using the NASA spin balance machine. Based on the EM spin balance pathfinder activities and preliminary calculations, balance correction was not expected following mating of the flight space vehicle assembly. Final unbalance measurements were well within the required specifications and balance correction to the processed space vehicle was not necessary.

Engineering Model Modal Survey

A modal survey was performed on the EM S/V to determine the global natural frequencies, modes shapes and damping coefficients. This data was used for the following purposes:

- To validate a Nastran model of the overall structure for coupled loads purposes
- To determine the in-orbit dynamics of the structure for assessing control loop interactions.
- Determine the local dynamic characteristics of key areas of the structure (panels, decks etc.)

A structural configuration was chosen for the test that would allow the primary vibration modes of the S/V to be measured while minimizing the influence of local modes and nonlinearities. In order to accomplish this, some components and subsystems were modified from the flight configuration. The following are examples:

- Deployable Solar Arrays - these were mostly non structural and mounted to the structure via hinges which would produce nonlinear responses.
- Sensor Bench - the mounting bench for the mission sensors was attached to the spacecraft via kinematic mounts. Although the sensor bench was left in for the test, an attempt was made to preload the mounts to reduce gapping.

- Propellant Tanks - Cylindrical steel mass simulators were used to simulate propellant filled tanks. The mounting flanges were flight like however.

The boundary conditions for the test were free-free with added interface mass. The free-free support system provides boundary conditions that are easily matched in an analytical model while the interface mass produces system mode shapes that have strain energy distributions that are similar to the modes see during launch. This was accomplished by mounting the S/V via the PLA to a 6 inch thick, 6 ft. X 12 ft. steel plate that weighed 15,000 lb. The whole system was then supported on a low frequency (less than 5 hz.) air bearing support system.

The modal shape vectors and frequencies from the modal survey were correlated with and used to refine the NASTRAN model; the final results for the first five modes are tabulated below.

Analytical Frequency (Hz)	Test Frequency (Hz)	Mode Shaped Description	Comments
21.3	23.7	Mid-deck Bending	Matched by axial random vibration test
32.7	33.8	Payload Bending, My	Good Match
35.2	38.2	Payload Bending, Mz	Non-linear test data due to friction slip at propellant tank mount
41.9	41.4	Sub-floor bending	Good Match
42.2	43.0	Delta V Motor Base Bending	Good Match

The mid-deck bending mode was correlated using data from the axial random vibration test because there was not sufficient energy input in the modal survey to break loose the sliding joints where the upper propellant tank bungs meet the middeck. These joints also caused some problems in one of the payload bending modes due to inconsistent slipping. This problem was solved in a later modal survey by locking the joints in place. Based on the results above, the analytical model was deemed acceptable for use in the coupled loads analysis.

Axial (X Axis) Vibro-Acoustic Test

Engineering Model Structure

A combined acoustic and thrust (X) axis random vibration test was performed on the Engineering Model Payload. The acoustic portion of the test simulated the acoustic noise environment seen during launch and Max Q while the mechanical vibration input simulated low frequency structure borne launch vibration. Since below 100 Hz the acoustic chamber is not capable of producing the required sound pressure levels, the mechanical vibration also served to fill in this area of the spectrum. The acoustic spectrum used for this test came from the Titan II users manual while the low frequency random vibration spectrum was an envelope of measured data from several different launch vehicles and had a bandwidth of 20 to 100 hz. The input test levels were 6 dB above the maximum expected flight levels for a two minute duration. These were considered qualification levels. For both structures, the low frequency random vibration input spectrum had to be tailored to prevent major structural components from exceeding design limit loads. The acoustic and random vibration input spectra are shown in figure 5.

The EM Payload was instrumented with accelerometers which were located in component mounting areas. Data from these were analyzed to provide random vibration PSD curves for evaluating and updating preliminary component random vibration specifications.

Flight Structure

A similar vibro-acoustic test was performed on the Flight Clementine S/V. The input spectra however, were lowered to 3 dB above maximum expected flight levels for a one minute duration. A subset of the accelerometers from the E/M test (seven) were again measured during this test. After scaling for input level differences and averaging the spectral data from each test, an interesting trend was obvious. Figure 6 compares the averages of the PSD measurements from the EM and the Flight Spacecraft. The Flight Spacecraft had significantly lower response at almost all frequencies above 100 hz. when compared to the E/M data. (Below 100 hz the response was driven by the electrodynamic shaker input and was not pure acoustic response) Averaging the Grms values of the same measurement points on the two tests again showed the E/M Grms response to be 4.1 dB higher than the Flight Spacecraft Grms response (7.01 vs. 4.35). It is likely that this is due to the substitution of flight components for mass simulators and the addition of thermal blanketing. It is assumed that the flight components, being assemblies of a generally complex nature, tend to absorb more energy than solid masses, especially at high frequencies. Also the thermal blankets may have significantly reduced the acoustic sound pressure impingement on the Flight structure. This information will be factored into preliminary vibro-acoustic predictions for future programs.

Component Random Vibration Testing

Acoustic sound pressure is generally the most significant driver of component random vibration. Early in the Clementine program a preliminary component random vibration test specification was predicted for component purchasing and early testing. This random vibration specification was estimated by scaling the vibro-acoustic response of similar types of spacecraft structures based on acoustic sound pressure input levels. After the E/M qualification vibro-acoustic test, the PSD responses that were measured were compared to the preliminary component random vibration spectrum. The responses were generally enveloped by the component spectrum, however there were narrow frequency bands where the responses exceeded the specification. Due to the inherent conservatism of shaker random vibration component vibration testing, these areas of exceedance were not deemed significant enough to warrant raising the specification.

Lateral Random Vibration

This test was originally intended as a system level workmanship / acceptance test for mechanisms and subsystem connections that did not get sufficiently tested during the axial axis combined vibro-acoustic test. However, after reviewing all of the EM test data it was decided that the axial vibro-acoustic test provided a good workmanship for the whole structure without any local over testing. Thus, lateral axis random vibration tests were not performed on the Flight Space Vehicle.

Quasi-static Loads Testing

During launch, ascent and trans-lunar phases of flight, the spacecraft is exposed to static loads and low frequency vibration (generally less than 50 Hz) due to rocket thrust, buffeting and other transient events. This is the most severe environment for the primary spacecraft structure and heavy components. These loads were applied to the Clementine Spacecraft using a 35,000 lbf. vibration shaker outputting sinusoidal excitation. The difficult aspect of this test was that the sinusoidal output of the shaker must be low enough in frequency so that minimal structural amplification occurs. Applying relatively high G levels at low frequency requires shaker armature displacements that can easily exceed the capability of the exciter (One inch Peak to Peak), so care had to be taken in choosing test parameters.

Quasi-static test loads were defined as 1.05 times design limit loads. Since this is a qualification test, it was performed on the EM S/C but not the Flight S/C. Before applying quasi-static loads, the DPA and ISA were removed and the S/C was mounted to the vibration fixture. The S/C was quasi-

statically tested alone because the shaker did not have enough force output to drive the full Space Vehicle Stack to the required loads. This was not a problem since the DPA and ISA were qualified in separate static loads tests.

The methodology for running this test was to use the sine vibration control software to command a sine dwell for one second at the desired amplitude and frequency. In reality, only one full cycle is required at full level but one second was the minimum allowable test time in the controller. Since this test we have modified the technique to set the over test abort limit to full test level. This causes a soft shutdown as soon as the shaker reaches full level and a reduction in the number of full level cycles in a 20 hz. test from 20 to about 3. Another refinement was to include an additional in-axis control accelerometer at the spacecraft center of gravity and use peak limiting control strategy. This provides an additional degree of control in the event of nonlinear structural behavior.

Quasi-static loads were applied in all three axes, the primary overall spacecraft loads were 12.0 Gpk acceleration in the axial direction and 3.5 Gpk in the lateral directions. The 12.0 Gpk axial load was input at 18 hz. There was some local amplification expected on the sensor bench and middeck due to the first bending mode of the middeck and a review of the post test data did show amplification of as much as 2.4 in these areas. However, these areas were designed for higher loads so this was not considered to be a problem.

The 3.5 Gpk lateral loads were input at 10 hz. No failures or significant structural degradation occurred during quasi-static testing. In addition to the quasi-static loads that were generated during launch and ascent, centrifugal loads were generated when the S/V was spinning at 60 RPM during its trans lunar stage of flight. These loads however, were covered during the spin balance test when the Space Vehicle is spun up to 80 RPM.

Pyroshock Testing

Engineering Model

A Pyroshock test was performed on the EM to assess the high frequency pyroshock environment and transmissibility. The structure was instrumented with shock accelerometers in component mounting areas. The weight of the SC was off loaded for this test. The Pyrotechnic shock events which occur in flight were initiated two times each during this test.

Data was collected from the accelerometers for each event and pyroshock response spectra were analyzed. These spectra were used for evaluating and updating preliminary component pyroshock specifications.

Flight Model

A Pyroshock test was performed on the Flight Space Vehicle Payload with Star Motor simulator in place of the Star Motor. This test provided a pyroshock acceptance test for the flight components. Acceleration response data was not recorded during this test. Pyrotechnic shock events which occur in flight were each initiated once.

Jitter Testing

A preliminary computer analysis indicated that during Clementine mission activities, the operation of certain on board mechanisms and equipment could cause vibration levels sufficient to impair the measurements made by the optical sensors. In order to assess the problem, testing was performed in which the Engineering Model Spacecraft was supported on a soft suspension system, the vibration sources

were activated or simulated and vibration responses were measured on the sensor bench. Two tri-axis accelerometers were mounted on each of the six sensor simulators, oriented so that sensor rotation could be measured. The vibration sources were simulated as follows:

- Solar Array Drive Mechanisms - The solar arrays were installed with bungee cords off loading the gravity force. The arrays were then rotated at all speeds that were feasible for mission operations.
- Reaction Wheels - The four reaction wheels were each run through their full range of operating speed.
- Inertial Measuring Unit (IMU) - One of the IMUs had a dither circuit that caused a 1000 hz. vibration output. The IMU was operated as in flight.
- The Attitude Control Thrusters - The attitude thrusters were a hydrazine mono propellant system that could not be operated in the laboratory. Based on past measurements, the force output of these is a square wave with the amplitude being the rated thruster force output. These forces were simulated using a modal survey type electrodynamic shaker.
- Cryo Cooler Compressor on the LWIR Sensor - The cooler on the LWIR sensor was mounted on the sensor bench with all of the optical sensors. The cooler compressor was operated as in flight.
- Mission scenarios were reviewed to determine which vibration sources might be in operation simultaneously. Two combined cases were identified and simulated. These were Lunar Mapping (cooler, IMU, reaction wheels and solar arrays) and Asteroid Flyby (thruster, cooler, IMU and solar arrays) These source combinations were then operated together.

The accelerometers on each sensor were measured simultaneously in order to maintain the phase relationship for accelerometers on the same sensor. As an example, figure 7 is a plot of the data measured on the UV Vis sensor due to operation of the solar array stepper motor at 200 hz. The top trace is the acceleration time history showing a peak of 50 milli-Gs. The bottom trace is the averaged auto spectrum of the same signal. For analysis purposes a worst case assumption was made, that being that the 50 milli-Gs was all concentrated at the lowest significant frequency of 67.5 hz.

These measured linear accelerations were then converted into angular accelerations based on geometric positions of the accelerometers. Angular velocities were then calculated by integrating the angular accelerations at critical frequencies. The angular velocities were compared directly to the sensor maximum allowable specification which was 1.0 micro-radian per millisecond. Incidentally, based on the data referenced above and shown in figure 7, 200 hz was not an acceptable speed for the solar array stepper motor. The following summarizes the conclusions that were reached:

- The two solar array rotation (stepper motor) speeds that caused the least jitter were found (100 hz - fast speed and 1 9/16 hz - slow speed). The mission parameters were modified to accommodate these.
- The responses produced by the reaction wheels were within acceptable limits for sensor operation.
- The IMU vibration was low level and did not cause a problem.
- ACS thruster pulse durations that produced minimum vibration response at the sensor bench were determined. The thrusters were one of the largest jitter sources.
- The Cryo Cooler Compressor operation produced a fundamental frequency of 47 hz but caused large harmonic acceleration responses on the sensor bench at higher frequencies. Integrating the

accelerations into velocities however, reduced the high frequency energy to the point where it was not a problem.

- The combined environment testing also showed acceptable results.

Summary

The Hybrid Qualification / Protoflight test concept worked well for this program. The Engineering Model testing produced valuable engineering data and insights into Clementine's structural integrity early in the program while cost and time saving were accomplished by only building flight units for many low risk components. Protoflight acceptance testing at 3 dB above flight levels was performed on all flight hardware, both at the component and system level. Some of the higher risk components, had non flight engineering models built which were qualification tested to 6 dB above flight levels early in the program to verify their design.

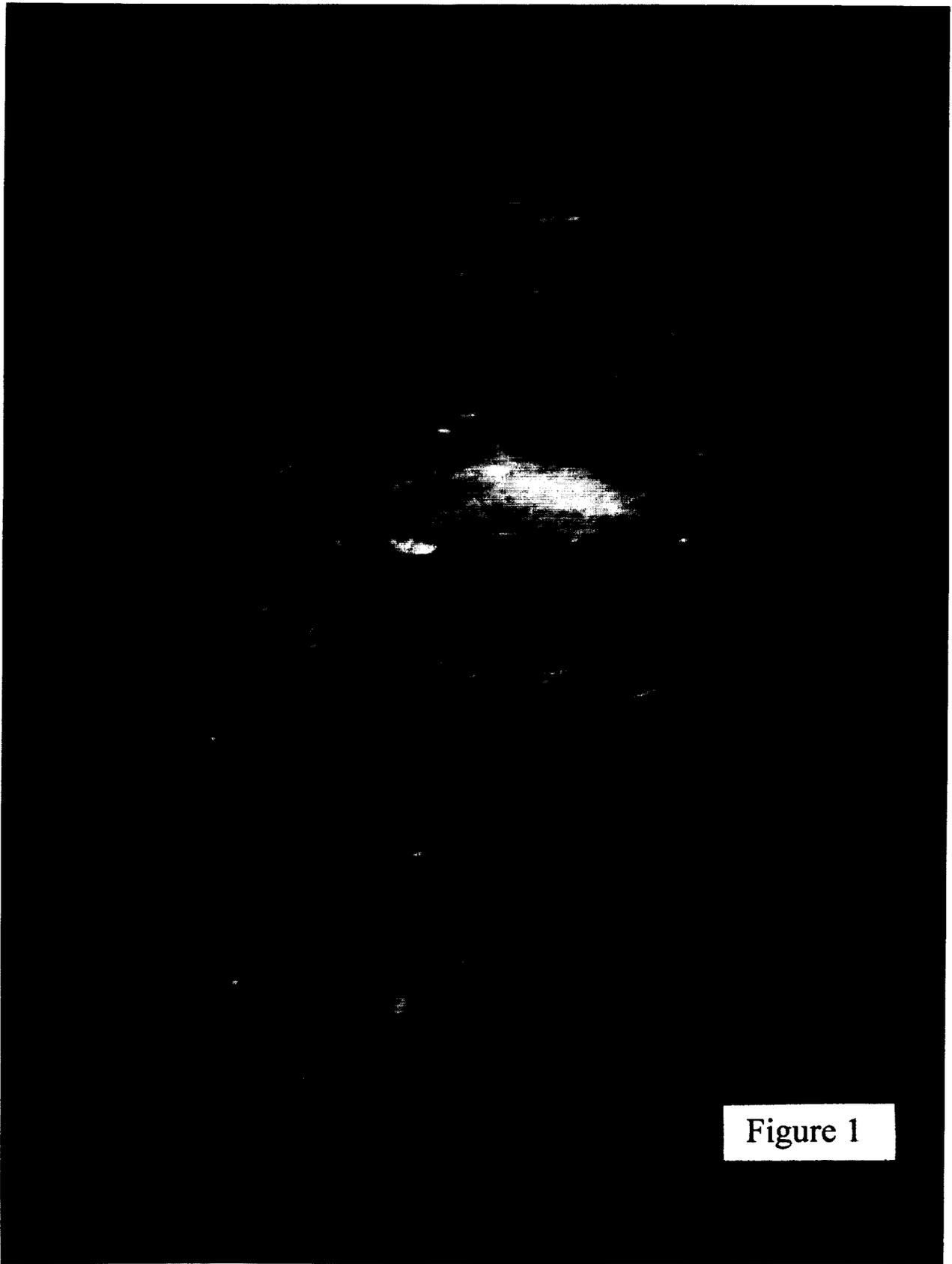
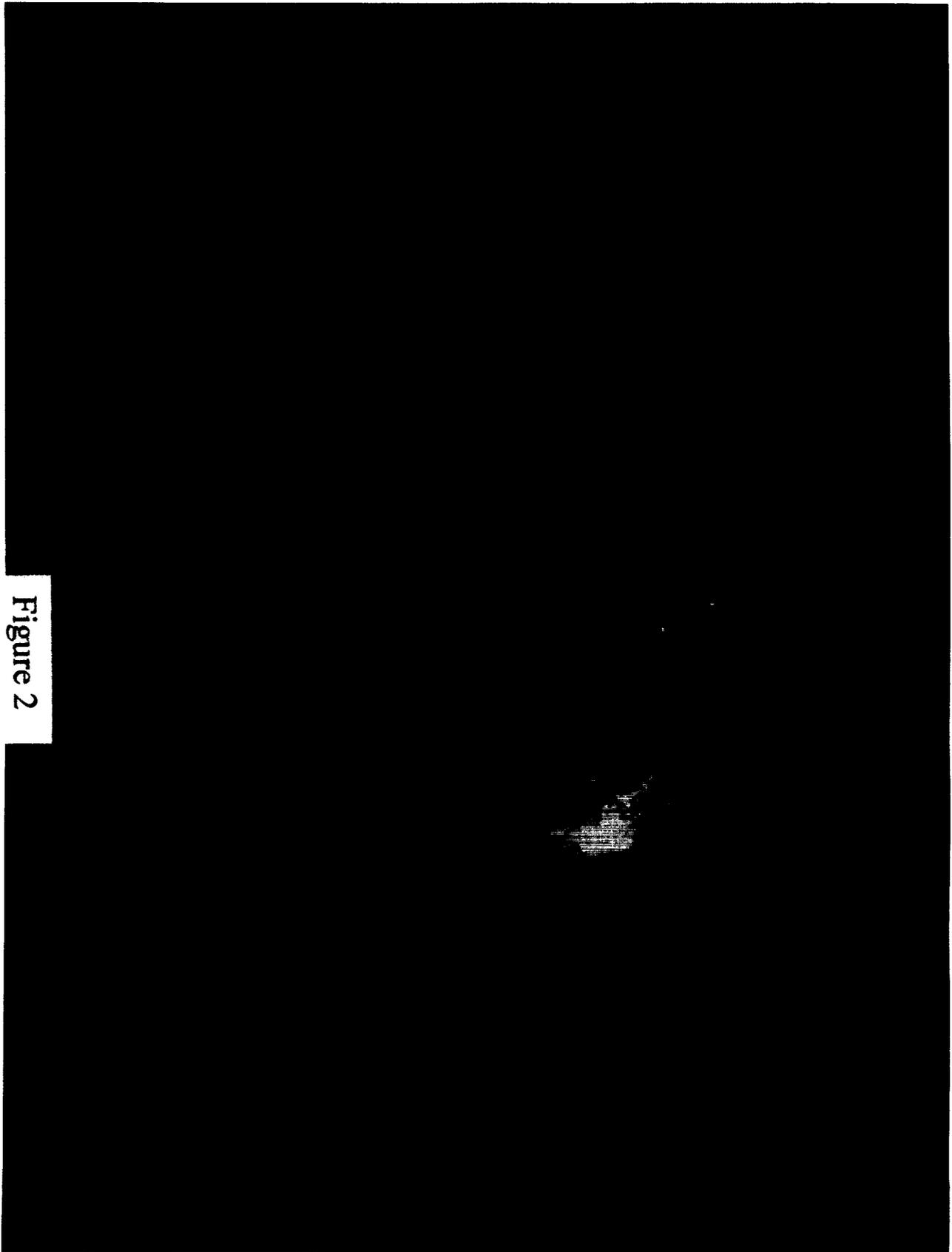


Figure 2



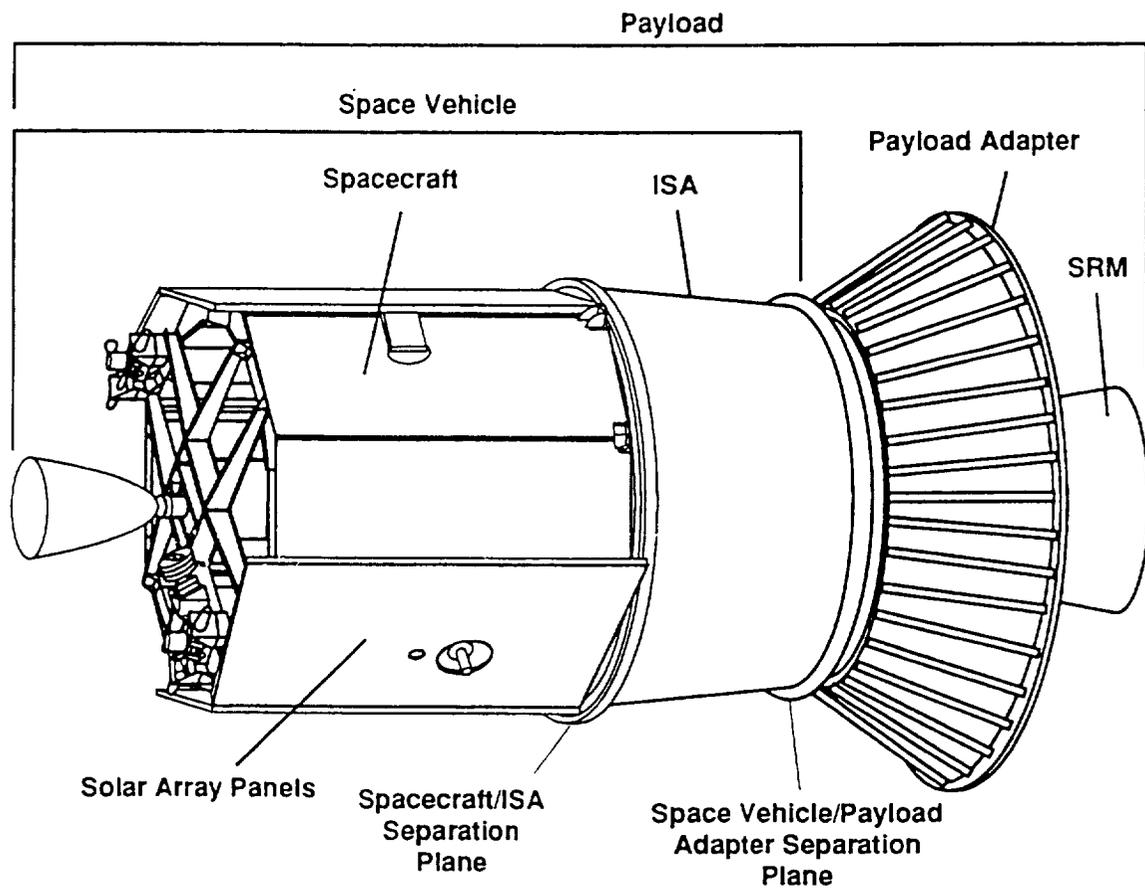
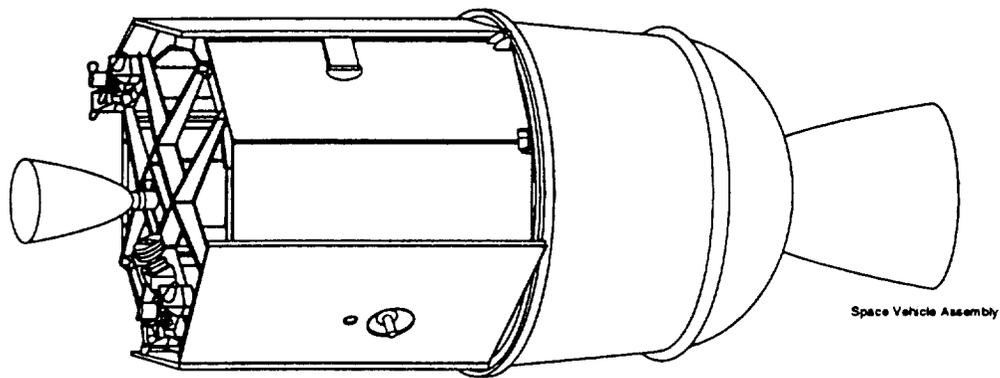
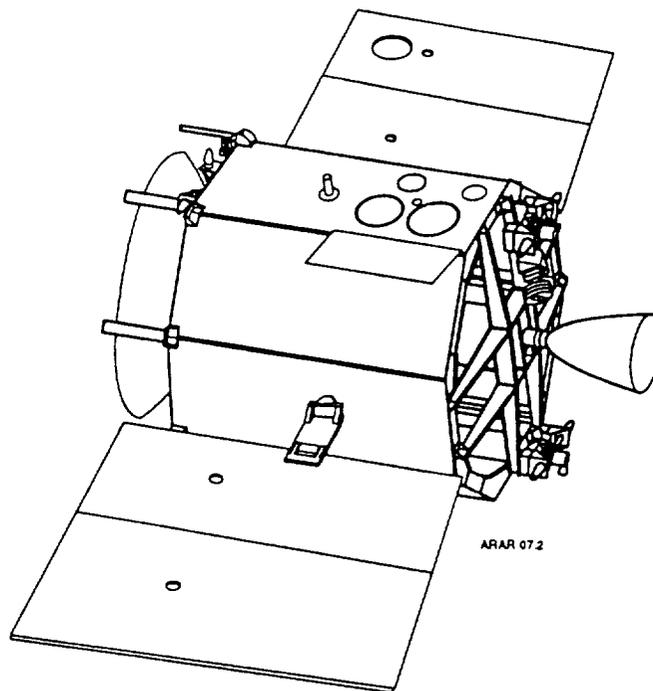


Figure 3

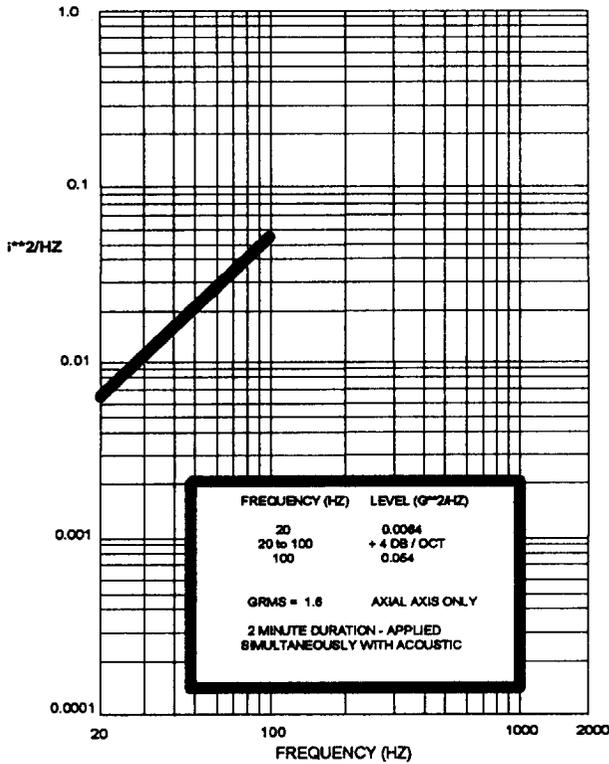


Configuration After Booster Separation

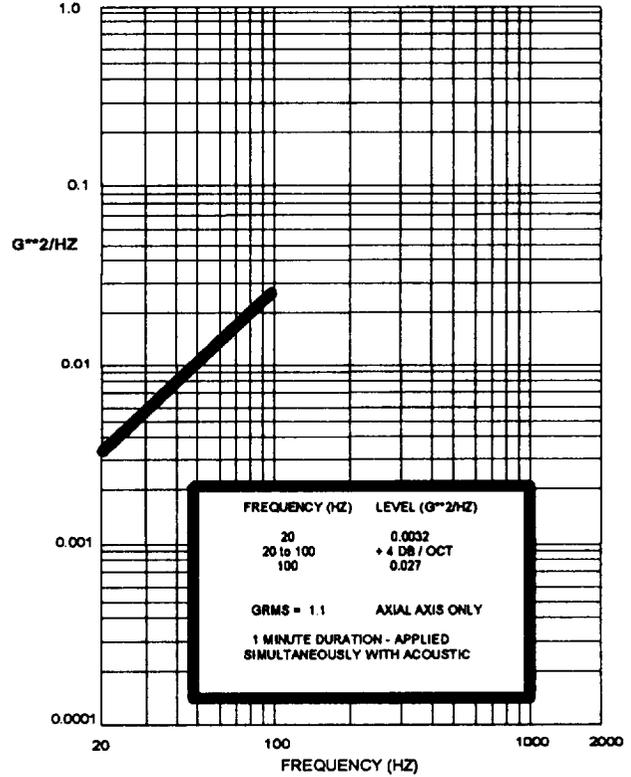


Configuration After Solid Rocket Motor Separation

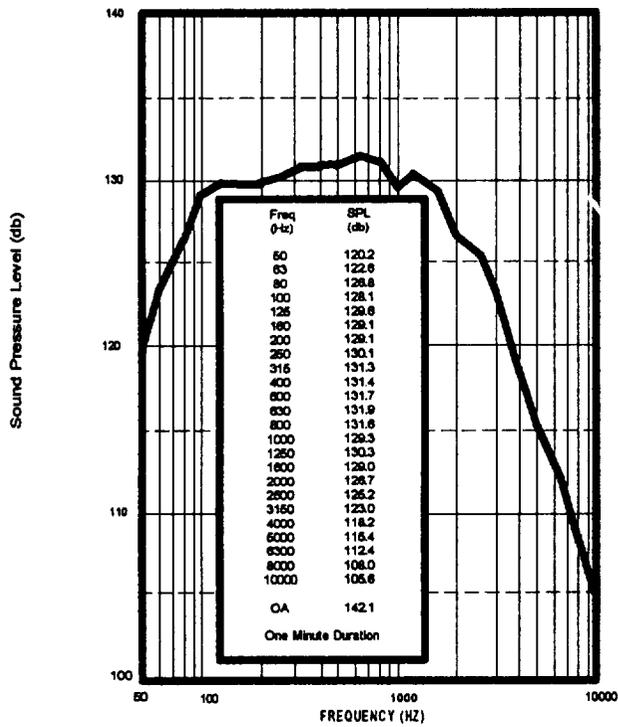
Figure 4



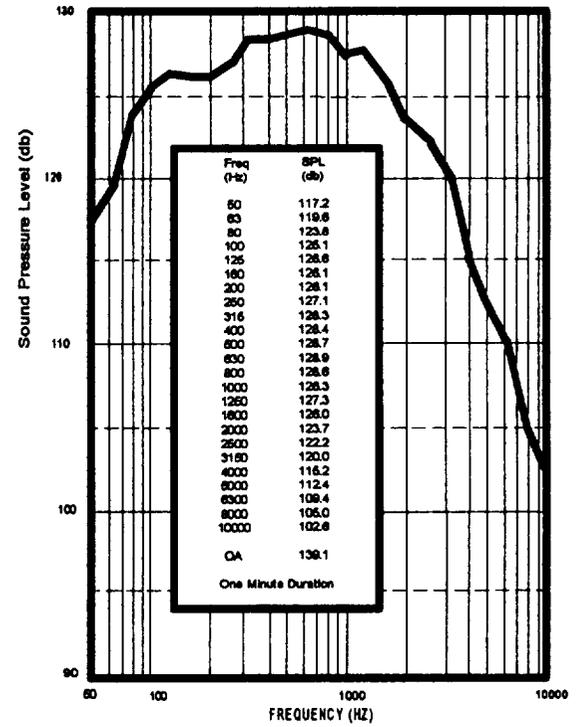
Qualification Random Vibration Spectrum for Engineering Model System Level Axial Vibro-acoustic Test



Protoflight Random Vibration Spectrum for Flight Space Vehicle System Level Axial Vibro-acoustic Test

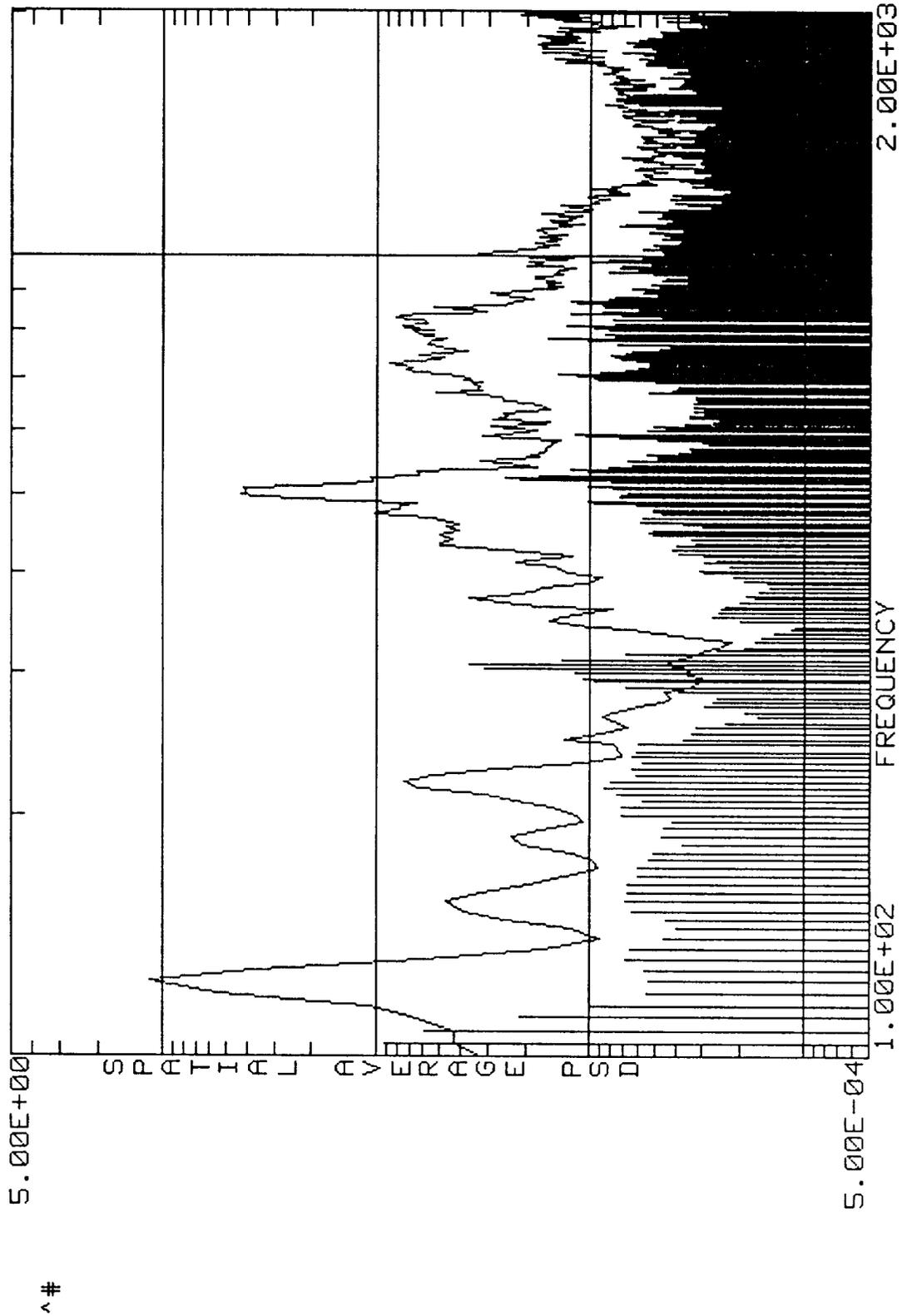


Qualification Acoustic Spectrum for Engineering Model System Level Axial Vibro-acoustic Test



Protoflight Acoustic Spectrum for Flight Space Vehicle for System Level Axial Vibro-acoustic Test

Figure 5



A2: LINE=E/M DATA
 A4: BARS=FLT S/C DATA

Figure 6

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INPUT 500Hz BaseBnd
AC AC >X
78mV 78mV .11V .88V
0 0 0 0
TRIGGER
Off

AVERAGING 111223
Add Ov1dReJ 234344
cnt= 40 **-----
WINDOWING
Hanning /a Normal
MODES
Frame=512 NonOverlp
AutoRng=Off Non-0pad

```

```

DISPLAY Double LOGx Hz
Cursor: Normal 1-Cur

STORAGE
F10= A200R2.dat
TS 20X 20Y-
Ov1File.dat
Playback.dat
rec= 0

Analog
OUTPUT Square 8Hz 200mV
Offset= 0 FilTon

RUN ENABLE HELP CONFIG
Off Off On Off

```

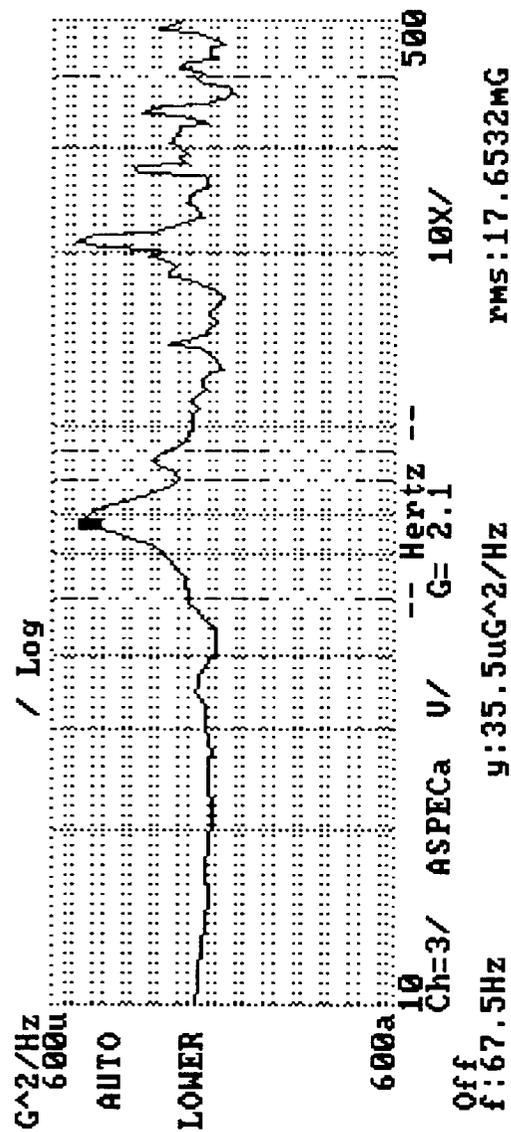
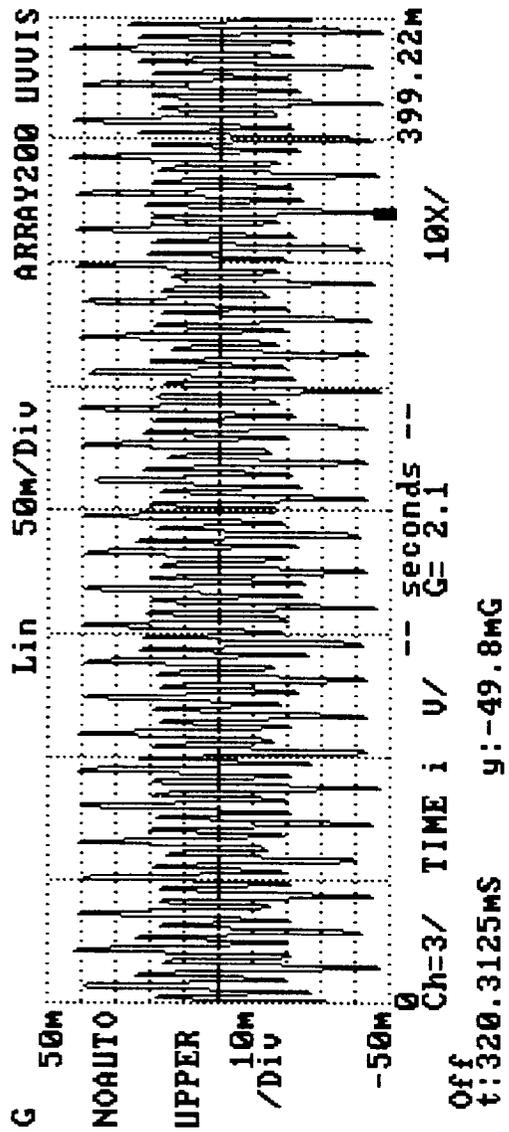


Figure 7

